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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

TECHNICAL NOTE 2311

FLIGHT INVESTIGATION OF THE VARIATION OF STATIC-PRESSURE ERROR OF A STATIC-PRESSURE TUBE WITH DISTANCE AHEAD OF A WING AND A FUSELAGE

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SUMMARY

The variation of static-pressure error with lift coefficient of a static-pressure tube located from 1/4 to 2 chords ahead of the wing of an airplane is presented. Similar calibrations of tubes located 1/2 to $1\frac{1}{2}$ body diameters ahead of the fuselage nose and 1 chord ahead of the wing of a second airplane are also presented.

The calibrations were determined by means of a trailing static tube over an indicated speed range from stall to speeds not exceeding 265 miles per hour. Each installation was calibrated in steady flight with the engine operating at rated power and with the flaps and landing gear retracted.

The tests of the wing-tip installations showed that the static-pressure error in the low-lift-coefficient range decreased progressively as the distance of the tube from the leading edge of the wing was increased from 1/4 to 2 chords. The error was shown to decrease quite rapidly at positions near the wing and at a lesser rate at greater distances. The error at lift coefficients near maximum lift coefficient, however, remained about the same for all positions of the tube.

The results of the tests of the fuselage-nose installations showed that the static-pressure error was reduced for all values of lift coefficient when the distance of the tube from the nose was increased. The change in static pressure error of the $1\frac{1}{2}$-diameter fuselage-nose installation over the lift-coefficient range was about the same as that for a 1-chord wing-tip installation on the same airplane.

INTRODUCTION

In designing an airspeed system for the research testing of an airplane, the primary problem is one of locating the static-pressure source in a region where the local static pressure will approximate stream static pressure for all values of lift coefficient and Mach number. A location which can be depended upon to meet this requirement for the greater part of the subsonic speed range is a position ahead of the wing tip (reference 1). A second installation, which may be equally satisfactory for subsonic operation and which is generally more satisfactory at transonic and supersonic speeds, is a position ahead of the fuselage nose. In either case, the problem becomes one of determining at what distance ahead of a given airplane configuration the static source should be located in order that the static-pressure error of the installation remain within specified limits.

In the case of wing-tip installations, the magnitude and variation of the static-pressure error with lift coefficient and Mach number depend on the distance of the static orifices ahead of the wing and on the shape and thickness ratio of the local airfoil section. For the usual wing section, the effects of airfoil shape and thickness ratio become comparatively small at a distance of about 1 chord ahead of the wing and this position has been generally adopted as a standard for research work. A collection of subsonic flight calibrations (unpublished) of 1-chord installations on a number of unswept-wing airplanes having a variety of airfoil sections with thickness ratios ranging from 7 to 12 has shown the static-pressure error to be small and a constant percent of the impact pressure at low lift coefficients. The errors at the high lift coefficients reached in the stall region, however, are usually undesirably large.

The static-pressure errors of fuselage-nose installations have been shown to depend on the distance of the static orifices ahead of the fuselage and on the shape of the nose section and the fineness ratio of the fuselage. The effect of nose shape has been determined from subsonic wind-tunnel tests of a static tube located various distances ahead of three bodies of revolution having hemispherical, ellipsoidal, and circular-arc nose shapes (reference 2). The effect of fineness ratio has been determined from unpublished wing-flow tests of a tube located various distances ahead of two bodies of revolution having circular-arc profiles of different fineness ratio. The results of these tests have shown that, for tube positions comparable to that of the usual 1-chord wing-tip installation, the magnitude of the static-pressure error is affected to an appreciable extent by the nose shape and fineness ratio of the fuselage.

Although the static-pressure errors of wing-tip installations have been fairly well established by tests on full-scale airplanes, the data were confined to only one position of the static-pressure tube. The errors of fuselage-nose installations, on the other hand, had been determined for a number of tube positions, but the data were restricted to tests of small-scale models without wings. It appeared desirable, therefore, to investigate various tube settings of a wing-tip installation with the primary objective of determining whether the errors at high lift coefficients might be reduced by extending the tube beyond 1 chord. Determination of the errors of fuselage-nose installations on a full-scale airplane to evaluate the effect, if any, of the wing-fuselage combination also seemed desirable. A series of flight tests has, therefore, been conducted to determine the variation of static-pressure error with lift coefficient of a static-pressure tube located various distances ahead of the wing tip of one airplane and at various distances ahead of the fuselage nose of a second airplane. A 1-chord wing-tip installation was also tested on the second airplane in order that a direct comparison of the characteristics of the two types of installation might be obtained. This paper presents the results of these investigations.

SYMBOLS

p	free-stream static pressure
p'	static pressure registered by pitot-static tube
p_T'	total pressure registered by pitot-static tube
Δp	static-pressure error $(p_r' - p_\alpha)$
q_c'	recorded impact pressure $\left(p_T' - p'\right)$
C_L	airplane lift coefficient
c	wing chord at spanwise location of static tube
t	maximum thickness of wing section at spanwise location of static tube
d	maximum diameter of fuselage
x	distance of orifices of static tube ahead of wing or fuselage

Subscript:

max maximum

APPARATUS AND TESTS

The type of pitot-static tube which was used for all the installations tested during this investigation is shown in figure 1. For the wing-tip-installation tests, the tube was located ahead of the wing of a trainer airplane by means of an adjustable boom which was attached to the underside of the wing parallel to the wing chord (fig. 2). The local chord at the spanwise station of the boom was 52 inches and the maximum thickness of the section at this point was 6.3 inches. The boom was set on successive flights with the static orifices of the tube at 1/4, 1/2, 3/4, 1, $1\frac{1}{2}$, and 2 chords ahead of the leading edge of the wing. For each setting of the boom, the static-pressure errors were determined by means of a trailing static-pressure tube (reference 3) suspended from the rear cockpit of the airplane. Calibrations were obtained over a speed range from stall to an indicated speed of about 240 miles per hour. Each installation was calibrated for the same flight condition, that is, rated power and with flaps and landing gear retracted.

The static-pressure errors were determined by measuring the differential pressure between the trailing tube and the static tube on the airplane. This differential pressure was recorded by an NACA differential-pressure recorder having a range of ± 2 inches of water. The trailing tube employed for these tests had a correction factor (established by wind-tunnel tests) of one-half of 1 percent of the impact pressure.

The fuselage-nose-installation tests were conducted on a fighter airplane equipped with an adjustable boom extending from the nose of the fuselage (fig. 3). A static-pressure tube was fitted to this boom and the static orifices set at 1 maximum fuselage diameter (56 in.) ahead of the nose. This installation was calibrated by means of a trailing static-pressure tube installed in one of the wing-tip fuel tanks. The NACA differential-pressure recorder used for this series of tests had a range of -1 to 6 inches of water. The tests were conducted over a speed range from stall to 265 miles per hour and the flight condition was the same as that for the trainer tests (rated power and with flaps and landing gear retracted).

The wing tanks were subsequently removed and a static-pressure tube was installed 1 chord ahead of the wing tip (fig. 3). The local chord at this station was 37.5 inches and the maximum thickness of the section 5.2 inches. The wing-tip installation was then calibrated against the 1-diameter fuselage-nose installation. On subsequent flights, the orifices of the tube on the fuselage boom were set at 1/2 and $1\frac{1}{2}$ diameters ahead of the nose and calibrated by comparison with the wing-tip installation.

RESULTS AND DISCUSSION

Wing-Tip Installations

Calibrations of a static-pressure tube located from 1/4 to 2 chords ahead of the leading edge of the wing of the trainer airplane are given in figures 4 to 9. Comparison of these figures shows that the variation of $\Delta p/q_c'$ with C_L for each of the installations is similar. In the low-lift-coefficient range the static-pressure errors are positive (above free-stream static pressure) and are more or less constant. At some higher lift coefficients the errors decrease to zero and, at still higher values of C_L, become increasingly negative as $C_{L_{max}}$ is approached. This variation of static pressure with lift coefficient has been previously shown to be characteristic of wing-tip installations (reference 4).

Further examination of these figures shows that the magnitudes of the static-pressure errors in the low-lift-coefficient range decrease progressively as the distance of the tube from the leading edge of the wing increases. The errors at lift coefficients near $C_{L_{max}}$, however, are of the same order for all positions of the static tube. At $C_L = 1.5$, for example, the error of each of the installations is about 5 percent q_c' below stream pressure.

The variation of static-pressure error in the low-lift-coefficient range with distance of the tube ahead of the wing is shown more clearly in figures 10(a) and 10(b). In figure 10(a), the static-pressure errors of the six installations at $C_L = 0.4$ have been plotted as a function of the position of the tube x expressed as a fraction of the local chord c. The static-pressure error decreases quite rapidly at positions near the wing and at a lesser rate at

greater distances. Increasing the boom length from $\frac{1}{4}$ to 1 chord, for example, reduces the error by 7 percent q_c', whereas a further increase to 2 chords results in an additional reduction of only 1/2 percent q_c'. As the magnitude of the static-pressure error ahead of a wing depends to a greater extent on the thickness of the local wing section than on the local chord, the static-pressure errors given in figure 10(a) have been replotted in figure 10(b) with the position of the tube expressed as a fraction of the maximum thickness t of the local wing section. As a means of indicating the general applicability of the results of the present tests at distances of the order of 10t, unpublished data on the static-pressure errors $\left(\text{at } C_L = 0.4\right)$ of 1-chord installations of nine other airplanes have been included in this figure. It will be noted that, for positions of from 8t to 12t, the static-pressure errors of the installations of these airplanes agree within 1 percent q_c' with the curve established by the tests of the trainer airplane.

Fuselage-Nose Installations

The calibrations of a static-pressure tube located 1/2, 1, and $1\frac{1}{2}$ body diameters ahead of the fuselage nose of the fighter airplane are presented in figure 11. The variation of $\Delta p/q_c'$ with C_L for the three installations is essentially the same, that is, the errors are positive and approximately constant in the low-lift-coefficient range and decrease slightly at the higher values of C_L. This variation of static-pressure error with lift coefficient is in general agreement with the results of wind-tunnel tests of reference 2, which showed that the static-pressure error at a given distance ahead of a body of revolution decreases with increasing angle of attack. The wind-tunnel data and flight data differ, however, as regards the magnitudes of the decrease at the three positions of the tube ahead of the body. The wind-tunnel data, for example, showed that the change in $\Delta p/q_c'$ due to angle of attack decreased as the distance x of the tube from the nose increased, whereas the flight tests show no consistent variation with tube position. As a matter of fact, the magnitude of the decreases for the $\frac{1}{2}$- and $1\frac{1}{2}$-diameter installations is exactly the same $\left(2 \text{ percent } q_c' \text{ for } C_L \text{ between 0.3 and 1.1}\right)$. That these differences are not due to differences in the static-pressure tubes used on the model and on the airplane may be seen from the fact that the errors given in reference 2 are actually position errors (installation error minus tube error) and the fact that the tube used on the

airplane is known to read correctly over the angle-of-attack range covered by the flight tests.

The differences between model and airplane results may rather be explained by the fact that the airplane installations are influenced by the pressure field of the wing as well as that of the fuselage. As the effect of both wing and fuselage is to cause the static-pressure error to decrease from positive to negative values as the angle of attack increases, it would be expected that the variation of $\Delta p/q_c'$ with C_L for the airplane installations would be greater than that for the model. The variations for the airplane installations, however, are less than those for the model and, as is shown subsequently, are about the same as those of a 1-chord installation on the wing tip of this same airplane. As the fuselage-nose installations are about 2 (root) chords ahead of the wing, the effect of the wing on these installations should (according to the wing-tip-installation tests) be slightly less than that for the 1-chord installation on the fighter airplane. Furthermore, as the nose of the fighter airplane is narrow in the transverse direction and would, therefore, produce less lift than a body of revolution, the greater part of the $\Delta p/q_c'$ variation of these particular nose installations would appear to be contributed by the wing rather than by the fuselage.

Figure 11 also shows that, in contrast to the calibrations of the wing-tip installations on the trainer airplane, the static-pressure errors at all values of C_L decrease as the distance of the tube ahead of the body is increased. At lift coefficients near the stall, for example, the errors of the three fuselage-nose installations decrease from 9 to 4 to 1 percent q_c', whereas the errors of the six wing-tip installations near $C_{L_{max}}$ were all about the same value (-5 percent q_c').

The manner in which the static-pressure errors in the low-lift-coefficient range vary with distance of the tube ahead of the fuselage nose is shown in figure 12. In this case, the static-pressure errors of the installations at $C_L = 0.4$ have been plotted as a function of the position of the tube x expressed in terms of the body diameter d. The error decreases from 11 to 5 to 3 percent q_c' as the position of the tube is increased from the 1/2 to $1\frac{1}{2}$ diameters. For comparison with the present tests, data from wind-tunnel tests of static-pressure

tubes located 1/4 to 2 body diameters ahead of a body of revolution (reference 2) have been included in this figure. The nose section of this wind-tunnel model was ellipsoidal and the data were obtained at a speed of 160 miles per hour and at an angle of attack of 0°. The data should, therefore, be approximately comparable to the low-lift-coefficient data obtained on the fighter airplane. The agreement between the two sets of data is probably as good as can be expected in view of the difference in nose shape of the model and the airplane.

Comparison of Wing-Tip and Fuselage-Nose Installations

The calibration of a static-pressure tube located 1 chord ahead of the wing tip of the fighter airplane is given in figure 13. The calibrations of the $1\frac{1}{2}$-diameter installation on the nose of this airplane and the 1-chord installation on the wing tip of the trainer airplane are also shown in this figure.

A comparison of the two wing-tip installations shows that the errors of the trainer installation are 1 percent q_c' higher than those of the fighter installation at low lift coefficients, although the errors of the two installations are about the same at higher values of C_L. The fact that the static-pressure errors of the trainer installation are higher than those of the fighter installation in the low-lift-coefficient range may be accounted for by differences in the pressure fields ahead of the two airplanes, for the airfoil sections of the two wings were quite different. The airfoil section of the fighter airplane, for example, is symmetrical with the point of maximum thickness at the 40-percent-chord station whereas that of the trainer airplane is cambered with the point of maximum thickness at the 30-percent-chord station.

A comparison of the calibrations of the 1-chord wing-tip and $1\frac{1}{2}$-diameter fuselage-nose installations on the fighter airplane shows that the magnitude of the static-pressure errors of the fuselage-nose installation ($x = 84$ in.) is considerably greater than that of the wing-tip installation ($x = 37.5$ in.). It must be remembered, however, that, in terms of the thickness factor, the position of the tube of the fuselage-nose installation is only 1.5, whereas that of the wing installation is about 7. Aside from the relative magnitudes of the errors of the two installations, it should be noted that the change in $\Delta p/q_c'$ over the entire lift-coefficient range is about the same for both installations.

CONCLUSIONS

From the results of low-subsonic flight tests of the static-pressure errors of a static-pressure tube located various distances ahead of a wing and a fuselage, the following conclusions have been reached:

1. The static-pressure errors of wing-tip installations in the low-lift-coefficient range decrease progressively as the distance of the tube from the leading edge of the wing increases. The decrease in error is greatest for positions near the wing and least for positions removed from the wing. Increasing the distance from 1/4 to 1 chord, for example, reduces the error by 7 percent of the impact pressure, whereas a further increase to 2 chords results in an additional reduction of only 1/2 percent.

2. The static-pressure errors of wing-tip installations near maximum lift coefficient are approximately the same (5 percent of the impact pressure below stream static pressure) for all positions of the tube between 1/4 and 2 chords.

3. The static-pressure errors of fuselage-nose installations are reduced throughout the entire lift-coefficient range when the distance of the tube from the fuselage nose is increased. In the low-lift-coefficient range the errors of a tube ahead of a fuselage having a nose shape approximating that of an ellipsoid decrease from 11 to 5 to 3 percent of the impact pressure when the tube is moved from 1/2 to 1 to $1\frac{1}{2}$ body diameters ahead of the nose.

4. A comparison of the calibrations of fuselage-nose and wing-tip installations showed that the change in static-pressure error over the lift-coefficient range was of the same order for the $1\frac{1}{2}$-diameter fuselage-nose installation and for a 1-chord wing-tip installation on the same airplane.

Langley Aeronautical Laboratory
National Advisory Committee for Aeronautics
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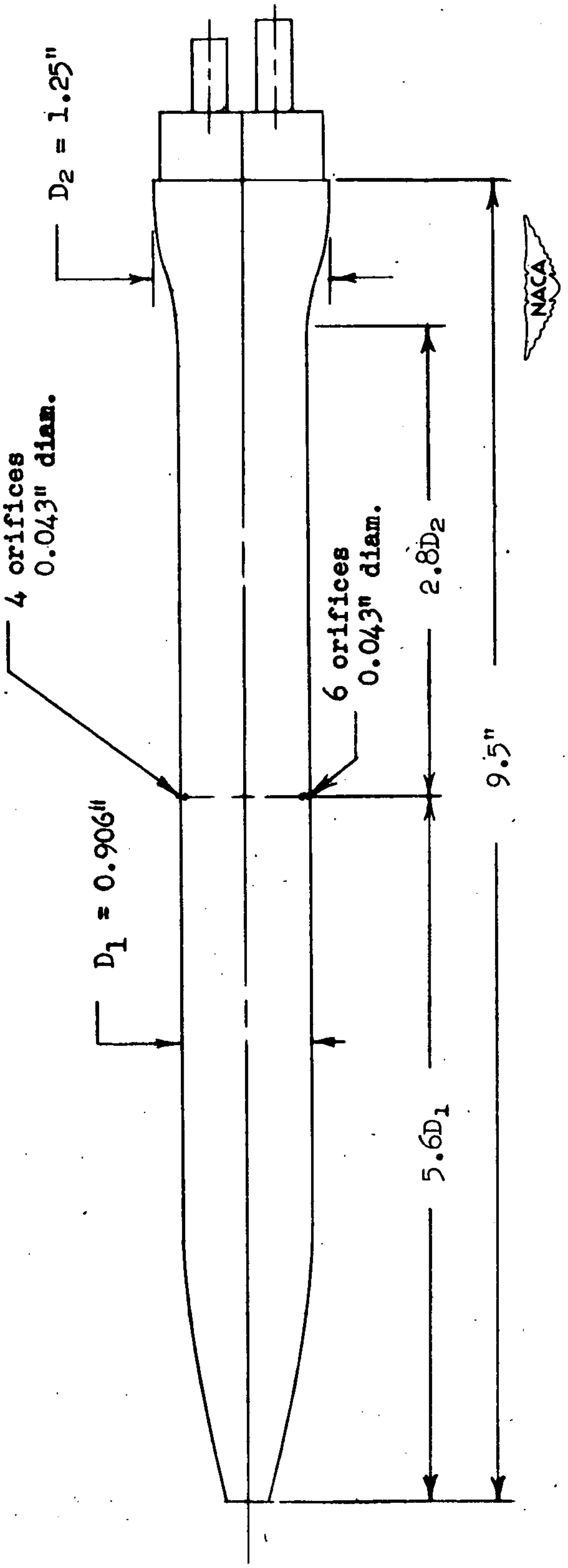

Figure 1.- Diagram of pitot-static tube used for static-pressure measurements.

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Figure 2.- Airspeed-tube installation on wing tip of trainer airplane.

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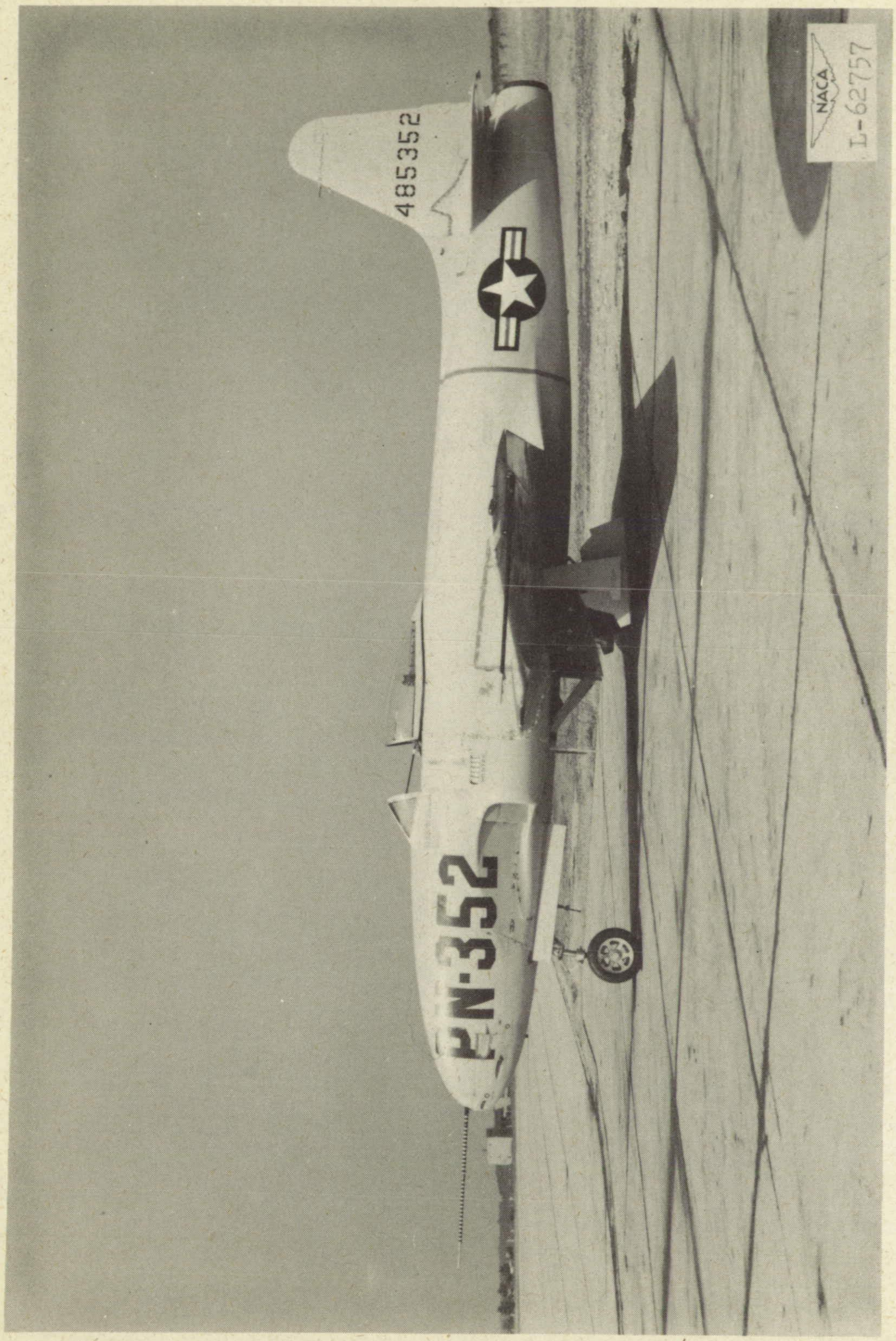

Figure 3.- Airspeed-tube installations on fuselage nose and wing tip of fighter airplane.

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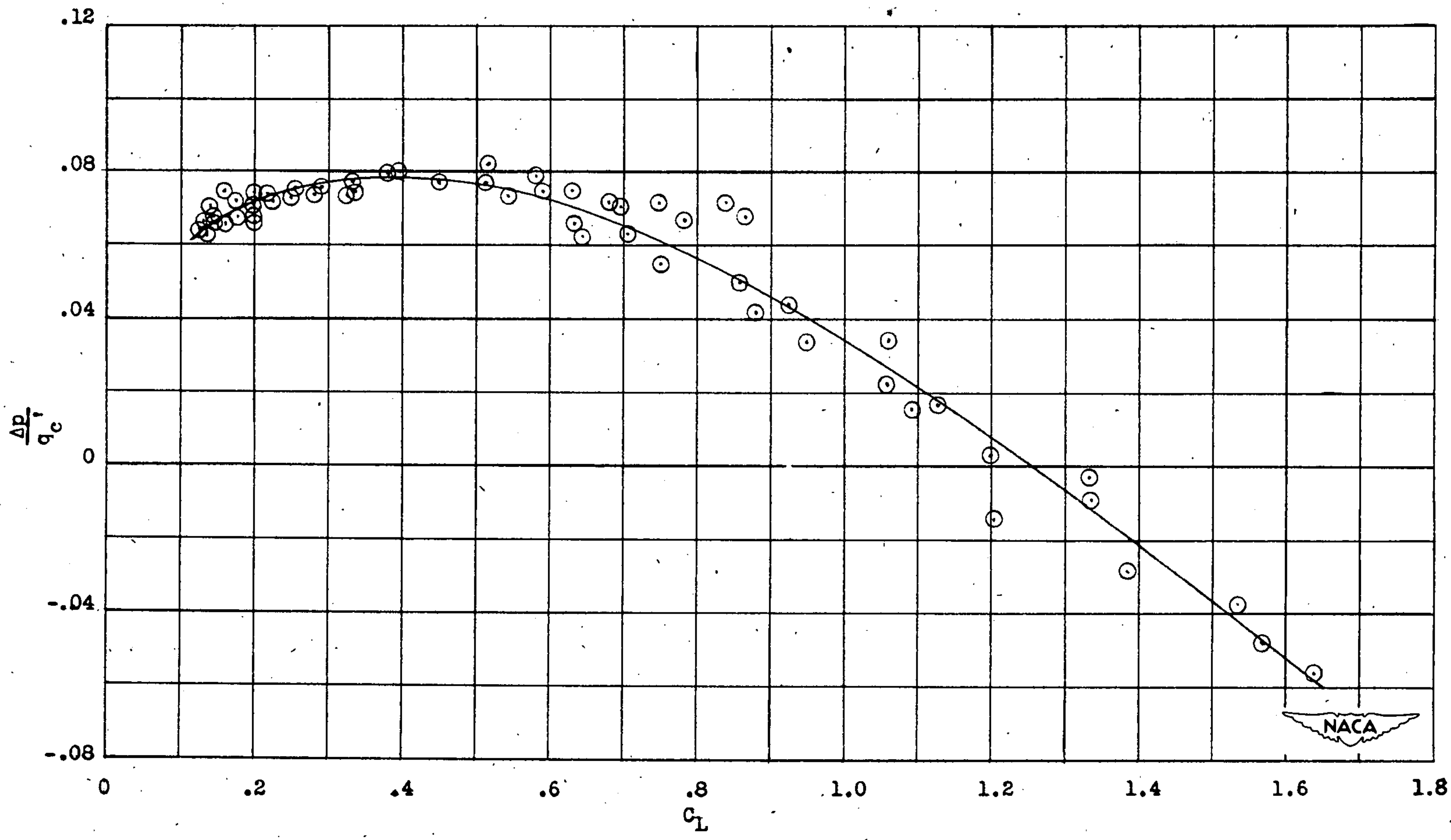


Figure 4.- Calibration of a static tube located 1/4 chord ahead of the wing tip of trainer airplane.

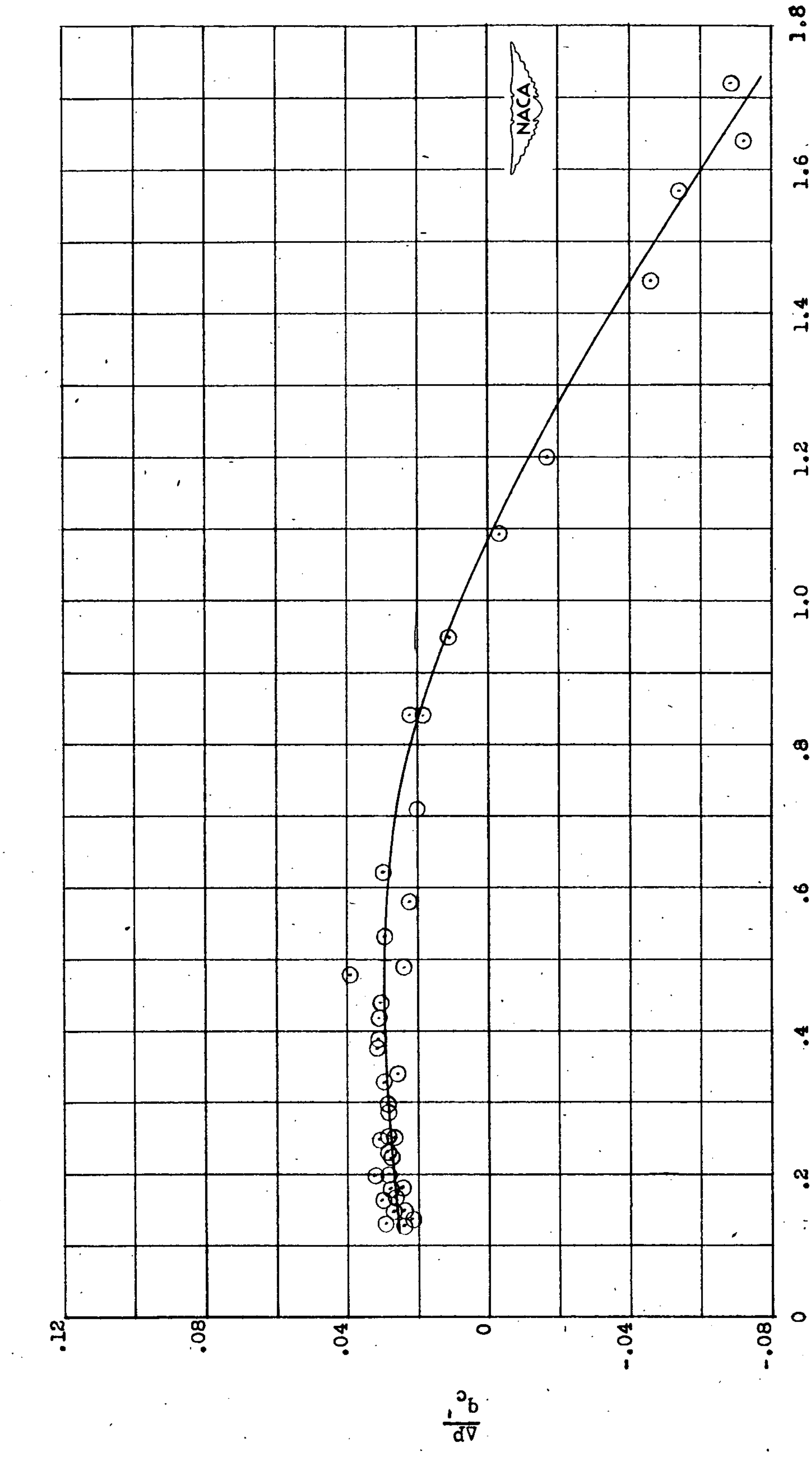


Figure 5.- Calibration of a static tube located 1/2 chord ahead of the wing tip of a trainer airplane.

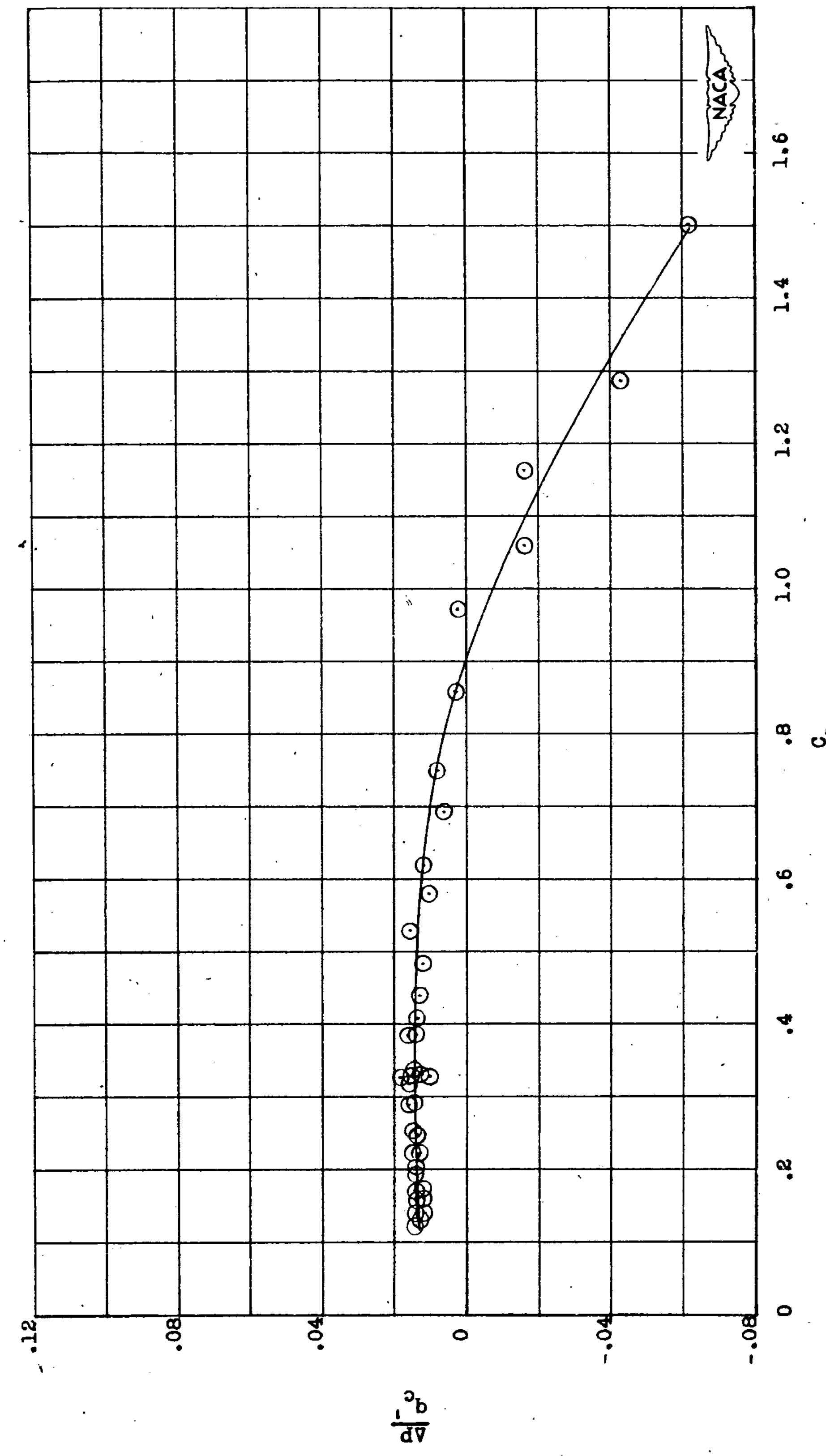


Figure 6.- Calibration of a static tube located 3/4 chord ahead of the wing tip of a trainer airplane.

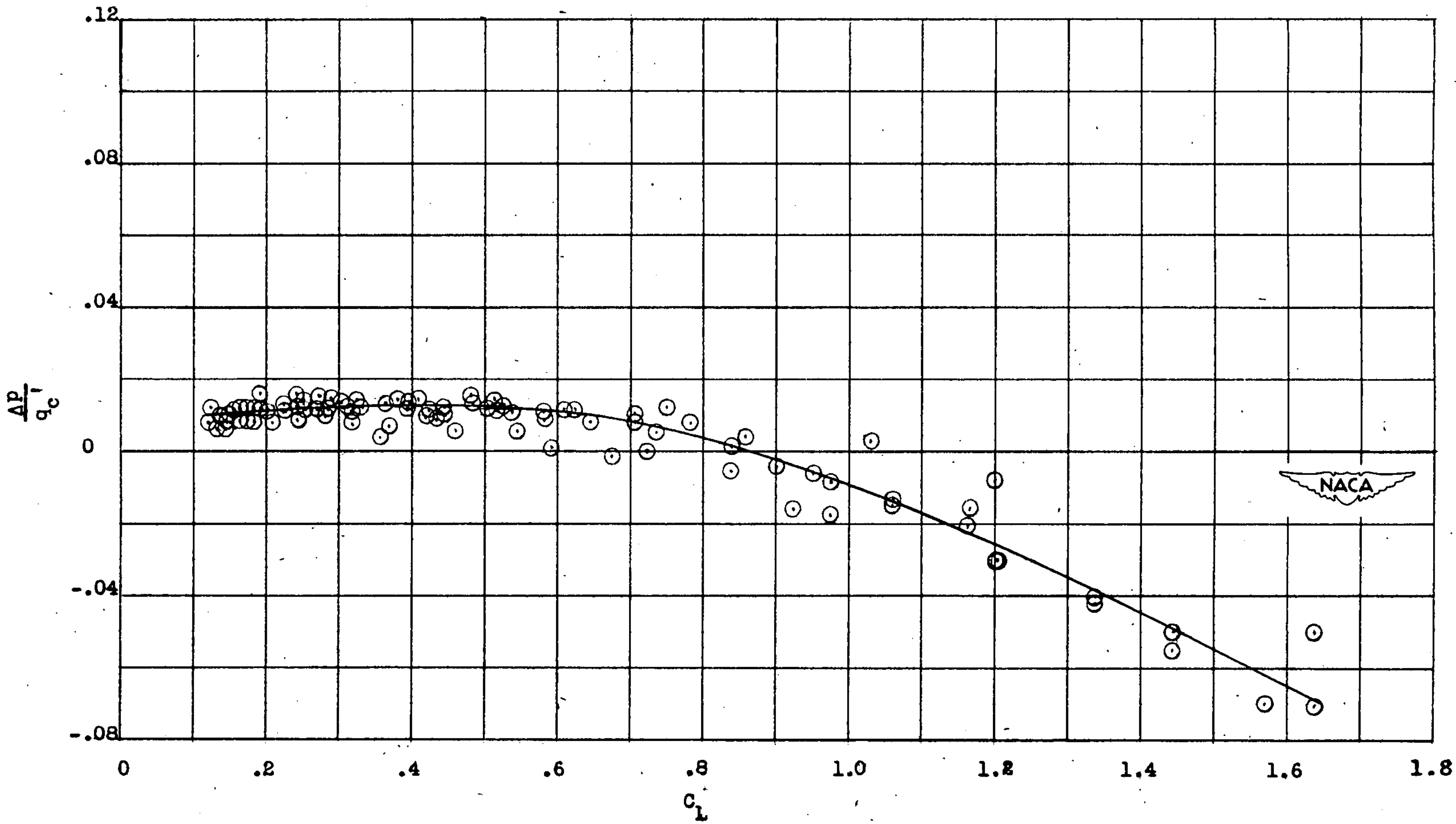


Figure 7.- Calibration of a static tube located 1 chord ahead of the wing tip of a trainer airplane.

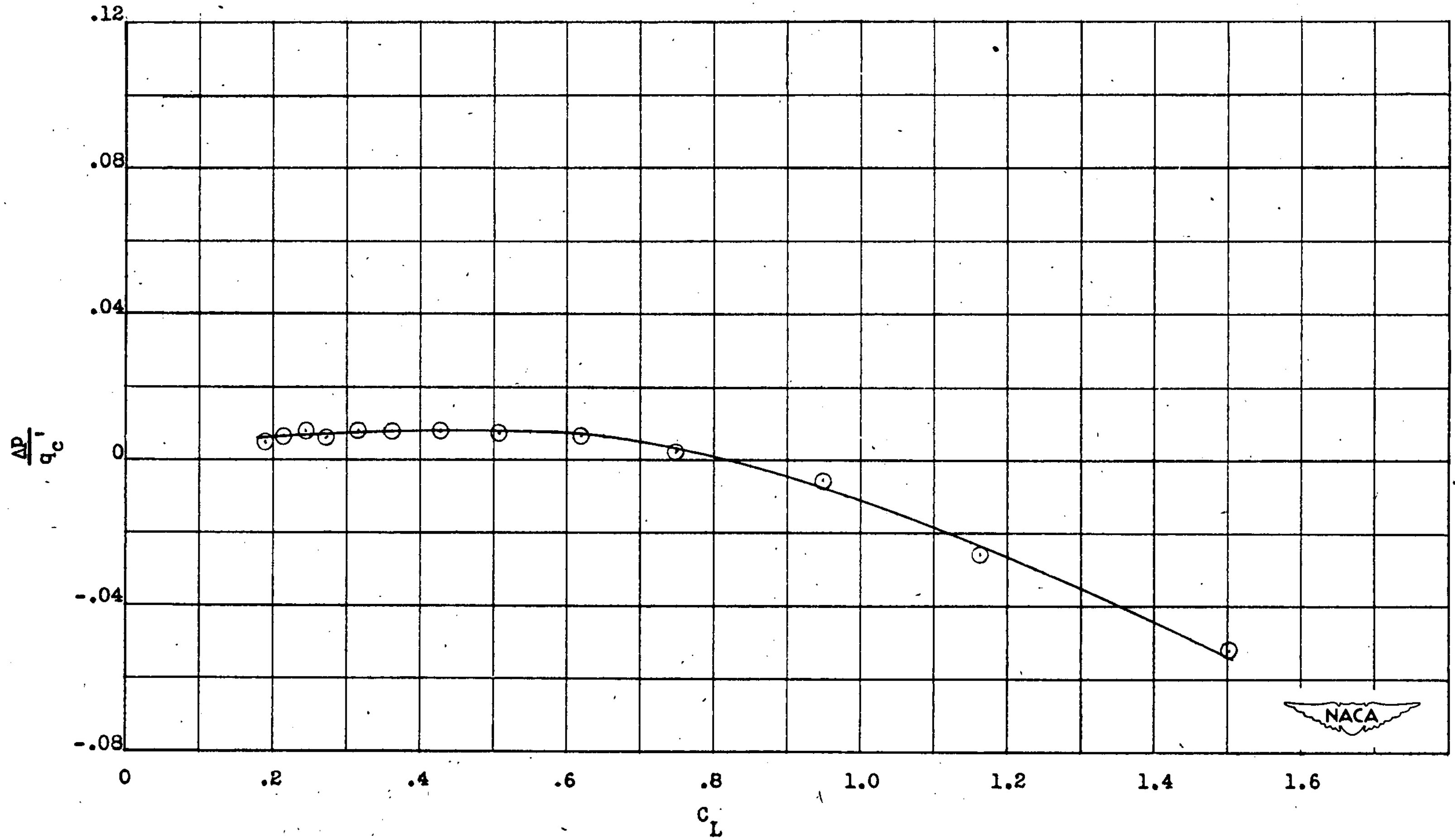


Figure 8.- Calibration of a static tube located $1\frac{1}{2}$ chords ahead of the wing tip of a trainer airplane.

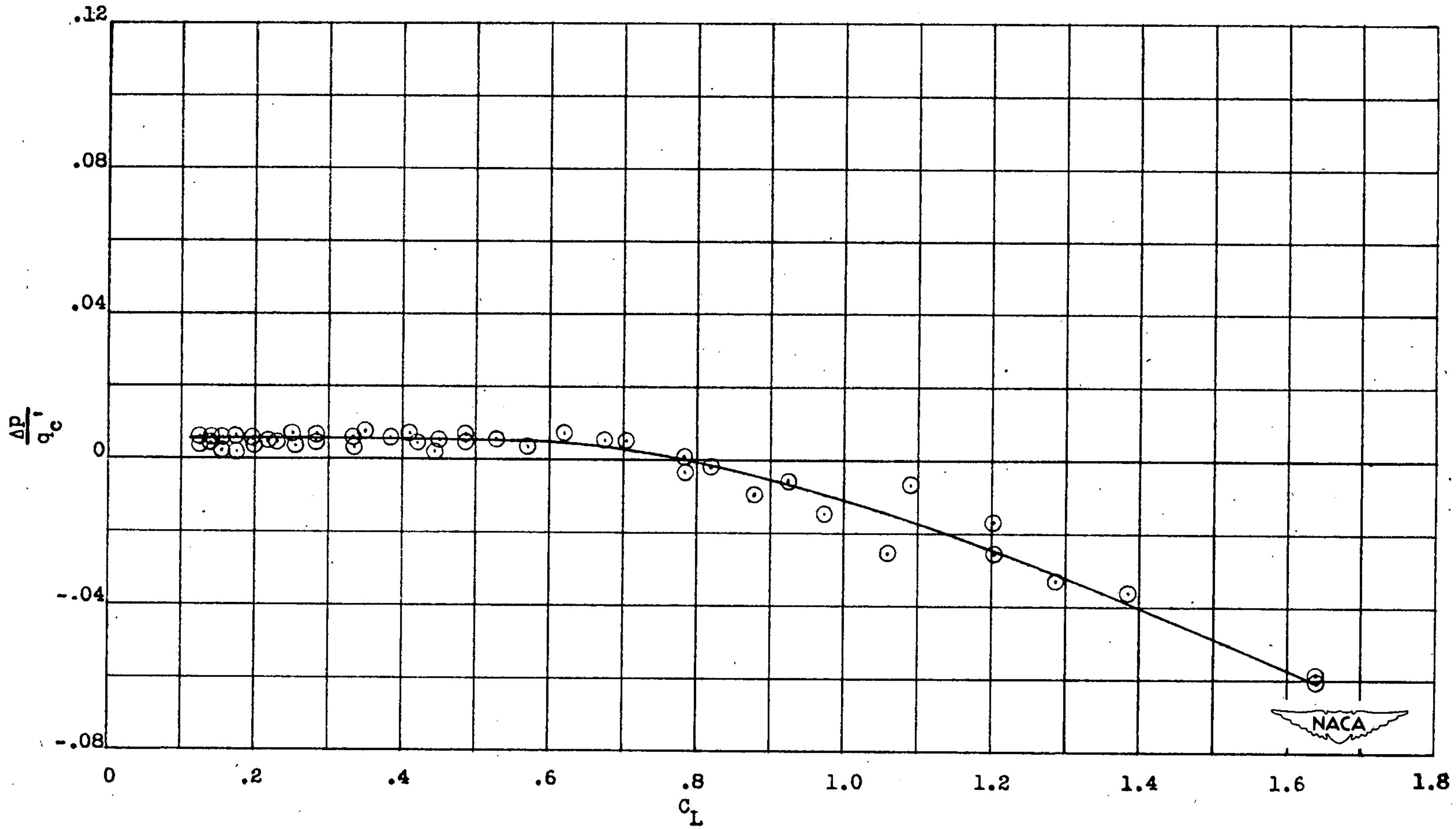
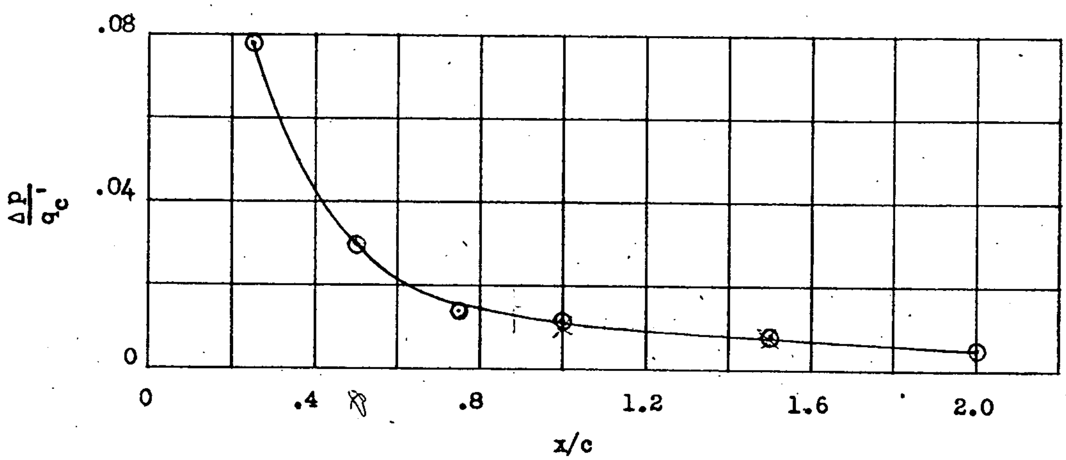
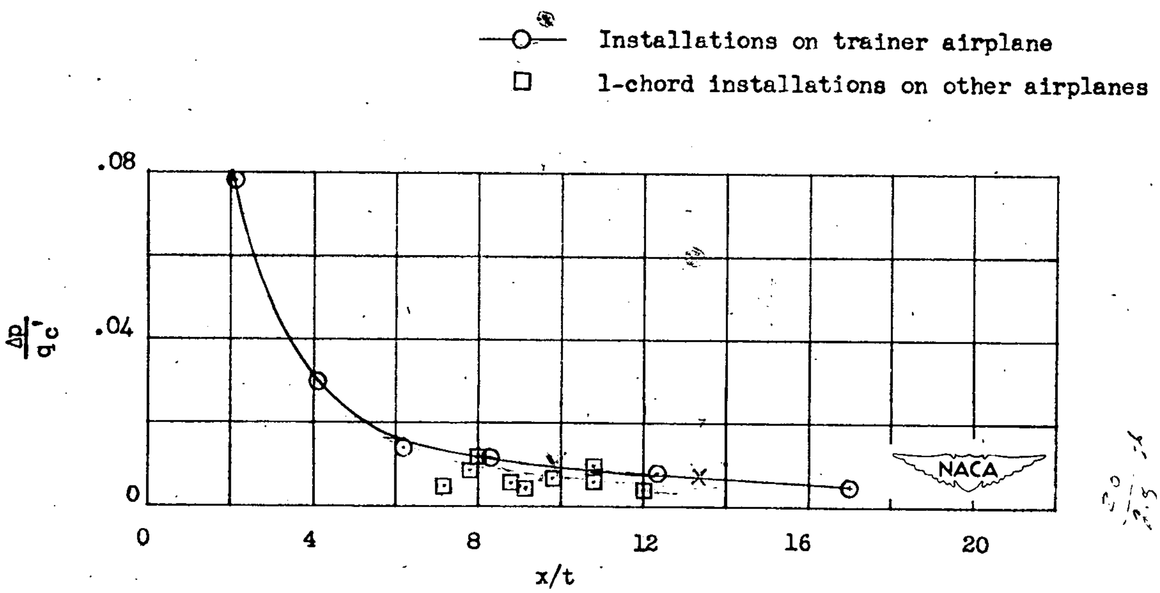


Figure 9.- Calibration of a static tube located 2 chords ahead of the wing tip of a trainer airplane.



(a) Orifice location as fraction of wing chord.



(b) Orifice location as fraction of maximum wing thickness.

Figure 10.- Variation of static-pressure error (at $C_L = 0.4$) with distance of tube orifices ahead of wing of trainer airplane.

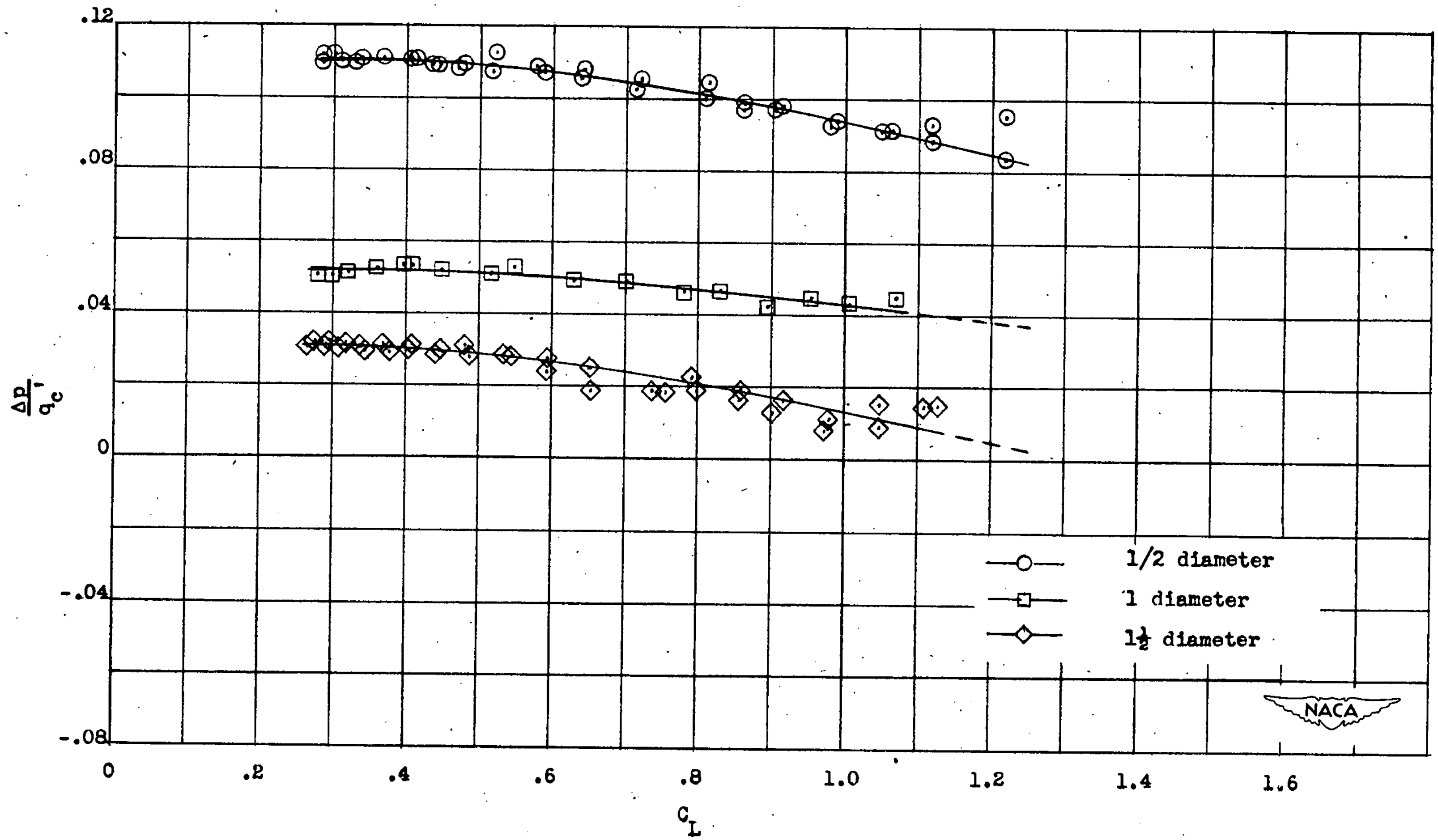


Figure 11.- Calibrations of a static tube located $\frac{1}{2}$, 1, and $1\frac{1}{2}$ body diameters ahead of the fuselage nose of a fighter airplane.

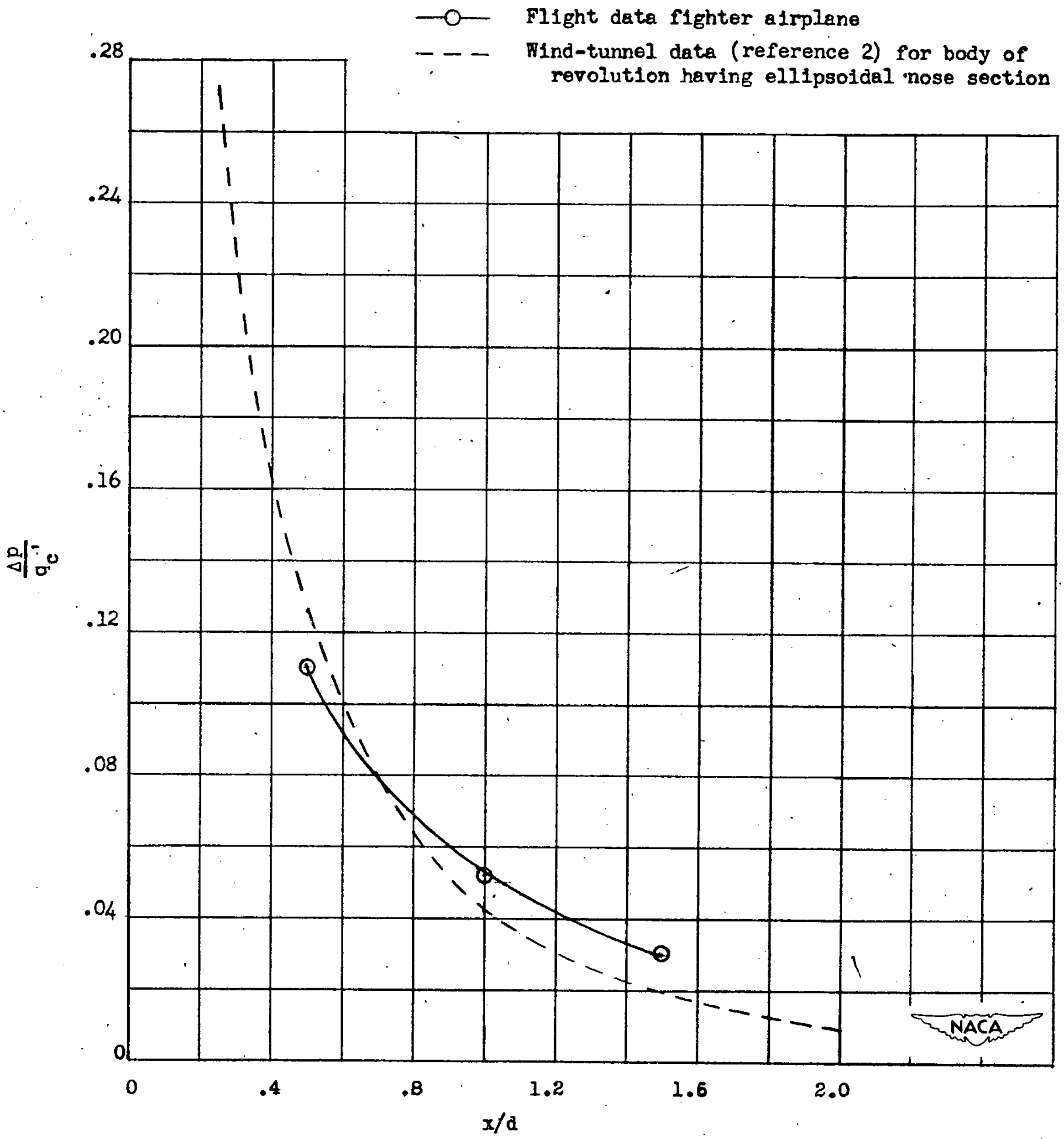


Figure 12.- Variation of static-pressure error (at $C_L = 0.4$) with distance of tube orifices ahead of fuselage of fighter airplane.

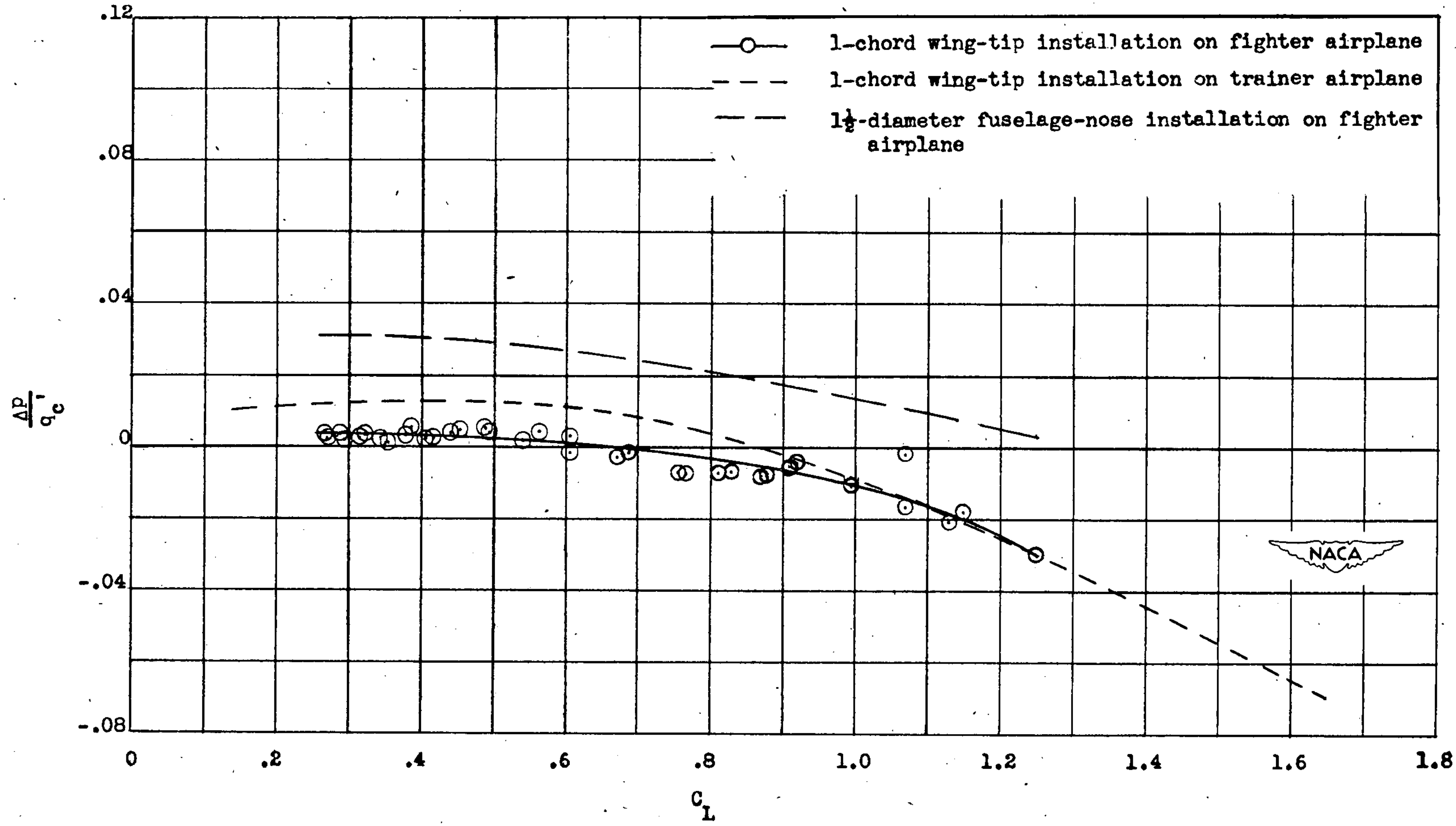


Figure 13.- Calibration of a static tube located 1 chord ahead of the wing tip of a fighter airplane.